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AIRCRAFT DYNAMICS AND LOADS COMPUTATIONS USING CFD METHODS

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Abstract

A novel approach to the art of computational simulation of modern flying vehicles using nonlinear aerodynamics by the method of CFD algorithms is presented and should hopefully advance the art of modern aircraft design. The method employs the concept of system identification to characterize nonlinear systems in terms of generalized state space coordinates. Furthermore, modal coordinates are used to formulate the equations of motion of the total vehicle system so that the computational effort will be kept at minimum cost. The proposed analytical concept is validated using simple test cases.

Keywords

Multidisciplinary optimization and design (MDO), Flight loads, Maneuver analysis, External loads.

Introduction

In recent years computational fluid dynamics (CFD) has evolved as a reliable aerodynamic loads prediction tool. But, its utilization in the design process is limited due to intensive computation required to obtain the air loads accounting for several variables such as structural flexibility, aircraft incidences and angular rates, flight control interaction and propulsion. An aircraft design which includes the interaction of multidisciplines would exhibit enhanced structural integrity and aircraft performance. This view point was expressed by the specialists of the multi-national cooperative program team who conducted several investigations to assess the impact of active controls technology (ACT) on the structural integrity of aerospace vehicles [1]. In a recent study Bhatia and Wertheimer have further emphasized the importance of aeroelastic effects in the design of high speed civil transport aircraft (HSCT) [2].

Towards this goal, a conceptual synergistic analysis methodology to compute flight loads and dynamic characteristics of air vehicles was presented in an earlier paper [3]. The multidisciplinary system considered in [3] consisted of the interaction of structures, inertia, digital flight control systems (DFCS), aerodynamics and propulsion systems. The concept of system identification was used to reduce the total

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vehicle system into a simplified state space model and to compute design loads.

As a natural extension, this paper discusses certain analysis and design issues associated with the application of the nonlinear airloads computed by the CFD methods. The CFD methods solve for airloads in time domain including the interaction of several variables such as angle of attack, control surface deflections, aircraft angular rates and accelerations and elastic deformation of the aircraft. Aircraft performance and stability analysis, and design methods often require the derivatives of the air loads with respect to state and design variables. Computation of these derivatives in a straight forward manner using CFD algorithms is computationally intensive. However, it is possible to estimate linearized derivatives of a nonlinear system by means of system identification concept. This paper presents an overview of the computation of the derivatives of nonlinear systems in an economical manner. These derivatives are then used to conduct flight maneuver analysis of aircraft. A brief outline of the approach is presented next.

Flight Maneuver Analysis

The dynamics of the aircraft consists of rigid body modes and vibration modes. The frequency spectrum of these modes are well separated, and hence, the rigid body equilibrium and elastic degrees of freedom can be solved separately. Thus, the rigid body equations of motion of maneuvering aircraft in wind axis system can be written as [3]:

$$\begin{Bmatrix} M\dot{V} \\ MV\dot{\gamma} \\ I_{yy}\dot{q} \\ \dot{\theta} \end{Bmatrix} = \begin{Bmatrix} F_x \\ F_z \\ M_y \\ 0 \end{Bmatrix} + \begin{Bmatrix} T\cos\alpha - Mg\sin\gamma \\ T\sin\alpha - Mg\cos\gamma \\ 0 \\ q \end{Bmatrix} \quad (1)$$

where M is the mass of the aircraft, I_{yy} is the pitching moment of inertia, V is the aircraft velocity along the flight path. The nonlinear aerodynamic forces as computed from the CFD code is given by the first column on the right hand side of equation (1). The body forces and thrust components are given by the first two rows of the second column. The climb angle denoted by γ and the pitch angle by, θ. The aerodynamic angle of attack is given by

$$\alpha = \theta - \gamma \quad (2)$$

The elastic equilibrium equations are given by

$$K_{ee}\eta_e = F_e \quad (3)$$

where K_{ee} is the generalized stiffness matrix, F_e is the generalized net load vector (i.e. sum of aerodynamic and inertia loads) corresponding to the vibration modes , η_e.

Equation (1) can be written in the state space form;

$$\dot{X} = F(X,u) \quad (4)$$

where the system level state space vector F accounts for synergistic contributions from various disciplines.

The state space vector, X is given by

$$X = \begin{Bmatrix} V \\ \gamma \\ q \\ \theta \end{Bmatrix} \quad (5)$$

and u denotes a vector of flight control variables such as pitch, roll, and yaw control commands, and including, if required, thrust vectoring control parameters.

Using truncated Taylor series, equation (4) can be rewritten as

$$\dot{X} = F(X_n) + \frac{\partial F}{\partial X}X + \frac{\partial F}{\partial u}u \quad (6)$$

The computation of the derivative matrices, $\frac{\partial F}{\partial X}$ and $\frac{\partial F}{\partial u}$ becomes expensive, especially if CFD methods are used to evaluate nonlinear aerodynamic loads. To overcome this difficulty and to simplify the design process in the manner of an automated sense, the present method employs the concept of system identification. Then, the equations of motion of a nonlinear system can be written in terms of state space coordinates. This approach has the potential to reduce the cost of design process and the design cycle time, in preliminary as well as in full scale design phases.

The aerodynamic solutions in CFD methods require large number of repeated computation of the flow variables. These solution steps can be used as the learning cycles to establish the dynamic characteristics of any nonlinear system using regression or neural network methods. These methods require a number of input and output samples taken at discrete time intervals. In any transient analysis it is possible to compute F and X at any time t_n. Then, using X as input and F as the output, the Jacobian of F at X_n and u_n can be computed.

Equation 6, together with equation 3, can be solved for the aircraft state variables, X, subject to the pilot command, u_c or the target load factor N_{ZT}. However, when dealing with agile aircraft which are designed with reduced stability margin, there is a need for feedback flight control laws to maintain aircraft stability and superior maneuver performances. A brief outline of the feedback control law design and transient maneuver analysis are presented in the following sections.

Feedback Control Law Design

The Hamiltonian cost function may be written as

$$H = \tfrac{1}{2}\varepsilon^T \mathrm{Q}\varepsilon + \tfrac{1}{2}u^T \mathrm{R}u + \lambda^T (F - \dot{X}) \qquad (7)$$

The first term is included to minimize the constraints, while the second term is used to minimize the control power requirement. The last term denotes the equations of motion with λ as a vector of Lagrangian constants.

For symmetric maneuver analysis the constraints are given by

$$\begin{array}{lll} \varepsilon(1) = N_Z - N_{ZT} & \text{normal load factor} & \\ \varepsilon(2) = \dot{\mathrm{q}} & \text{pitch acceleration} & \end{array} \qquad (8)$$

The normal acceleration at aircraft center of gravity is given by

$$a_{Z_{cg}} = V\dot{\gamma} \qquad (9)$$

Then, the corresponding load factor is

$$N_Z = \frac{V\dot{\gamma}}{g} \qquad (10)$$

The error function ε, using equations (6) and (8) can be written as

$$\varepsilon = \mathrm{C}X + \mathrm{D}u + f_0 - f_T \qquad (11)$$

where the target vector quantity is given by

$$f_T = \begin{Bmatrix} N_{ZT} \\ 0 \end{Bmatrix} \qquad (12)$$

Since, the target acceleration is zero, the maneuver represents steady pull-up or push-down pitch rates depending on the sign of the load factor, N_{ZT}. The desired final condition can be achieved by means of optimally selected control input, u, as discussed in the following paragraphs.

The principle of optimal control theory yields the following Hamiltonian matrix equations (i.e. state and costate equations);

$$\begin{Bmatrix} \dot{X} \\ \dot{\lambda} \end{Bmatrix} = \begin{bmatrix} A & -B \\ -Q & -A^T \end{bmatrix} \begin{Bmatrix} X \\ \lambda \end{Bmatrix} + \begin{bmatrix} F_1 \\ F_2 \end{bmatrix} \qquad (13)$$

together with the necessary feedback control

$$u = -\Re^{-1}[\hat{\mathrm{Q}}^{\mathrm{T}} X + \frac{\partial F}{\partial u}\lambda + f'] \qquad (14)$$

where,

$$A = \frac{\partial F}{\partial X} - \frac{\partial F}{\partial u}\Re^{-1}\hat{Q}^T \qquad (15)$$

$$B = \frac{\partial F}{\partial u}\Re^{-1}\frac{\partial F}{\partial u}^T \qquad (16)$$

$$Q = \overline{\mathrm{Q}} - \hat{\mathrm{Q}}\Re^{-1}\hat{\mathrm{Q}}^{\mathrm{T}} \qquad (17)$$

in which

$$\overline{\mathrm{Q}} = \mathrm{C}^{\mathrm{T}}\mathrm{QC} \qquad (18)$$

$$\hat{Q} = C^T Q D \tag{19}$$

$$\Re = R + D^T Q D \tag{20}$$

$$f = f_0 - f_T \tag{21}$$

$$f' = D^T Q f \tag{22}$$

$$F_1 = F_0 + \frac{\partial F}{\partial u}\Re^{-1} f' \tag{23}$$

$$F_2 = C^T Q f - D\Re^{-1} f' \tag{24}$$

The solution of the two point boundary value problem of equation (13) can be written as;

$$\begin{aligned}\xi(t) &= \Phi(t,t_i)\xi(t_i) + \Psi(t,t_i)\begin{Bmatrix} F_1 \\ F_2 \end{Bmatrix} \\ &= \Phi(t,t_i)\xi(t_i) + \begin{Bmatrix} f_1 \\ f_2 \end{Bmatrix}\end{aligned} \tag{25}$$

where

$$\xi = \begin{Bmatrix} X \\ \lambda \end{Bmatrix} \tag{26}$$

and, Φ and Ψ denote transition matrices. The end conditions are given by

$$X(t_0) = X_0 \tag{27}$$

and

$$\lambda(t_f) = Q_f \varepsilon(t_f) = 0 \tag{28}$$

because the error at target time is zero. Hence, Lagrangian coefficients are given by

$$\lambda = -\Phi_{22}^{-1}[\Phi_{21} X + f_2] \tag{29}$$

Finally, the feedback control can be written as

$$u = KX + s \tag{30}$$

where

$$K = -\Re^{-1}[\hat{Q}^T - \frac{\partial F^T}{\partial u}\Phi_{22}^{-1}\Phi_{21}] \tag{31}$$

and

$$s = -\Re^{-1}[D^T Q N_{ZT}' + \frac{\partial F^T}{\partial u}\Phi_{22}^{-1} f_2] \tag{32}$$

This paper seeks to address two problems. The first problem is, the computation of maneuver design loads including the interaction of flight control systems. The solution to this problem is given by equation (25).

The second problem is the prediction of aeroelastic instability, such as flutter and divergence. The aeroelastic stability characteristics of a nonlinear system can be determined by means of the Lyapunov energy criteria, which in the simplest form, may be stated as;

$$V = \frac{1}{2} X^T P X > 0 \tag{33}$$

and its rate is given by

$$\dot{V} = \frac{1}{2} F^T P X < 0 \tag{34}$$

where P is the Riccati matrix which is derived from the eigenvectors of the Hamiltonian matrix of equation (13). The elements of the Hamiltonian matrix are denoted by A, B, and Q matrices.

Flight Maneuver Module :

ENSAERO CFD code [5 and 6], developed by NASA/Ames, has been updated with a flight maneuver analysis module. This module computes the generalized force vector, F, and solves for the aircraft state vector, X. The local derivatives of the of the nonlinear generalized force vector, $\frac{\partial F}{\partial X}$ and $\frac{\partial F}{\partial u}$. are computed using a modified version of the autoregression algorithm. These derivatives are used to compute the feedback control input, u. A subroutine to compute the Riccati matrix, P, and the Lyapunov functions V and $\dot{V}$, is also included in the flight maneuver module to compute static and aeroelastic stability characteristics (flutter and divergence speeds) of the aircraft.

Discussion of the Results :

The double delta wing configuration shown in Figure 1 is used to compute unsteady airloads and flight maneuver loads including its aeroelastic stability. The wing has a circular airfoil of 6% thick, and leading edge sweep angle of 50.4 degrees. The control surface is located at 56.6% span and behind 80% chord line. This wing was tested in the NASA Langley's transonic dynamic tunnel (TDT). The steady and unsteady airload data are reported in Reference [8]. The preliminary results are reported in the present version of this paper. The final paper will include stability derivatives and the results of maneuver loads analysis.

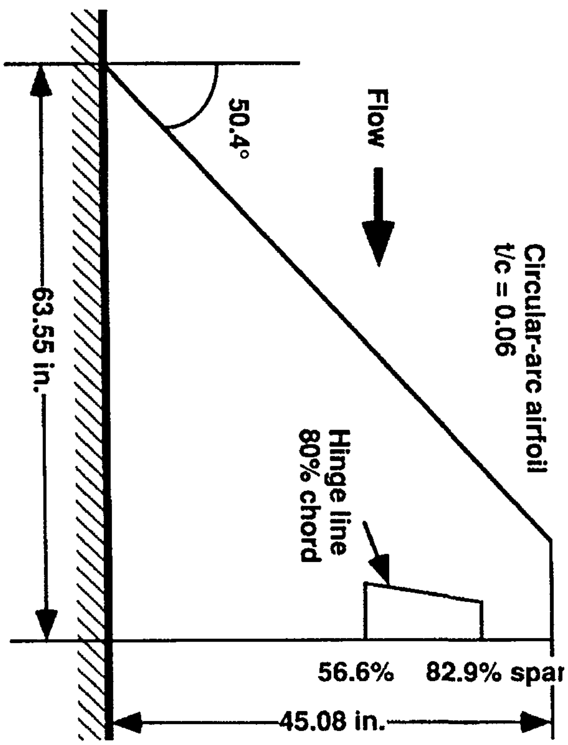

Figure 1. Planform of Clipped Delta Wing with Trailing-Edge Control Surface

Unsteady Airloads Computation :

The flow domain of this wing was represented by array of C-H grids consisting of 151 nodes in the flow direction, 44 nodes in the span direction and 34 nodes in the normal direction to the wing planform. Baldwin-Lomax viscous model was used to capture the leading edge vortex phenomena. To verify the accuracy of the ENSAERO code in oscillatory motion the control surface motion was set at 8 Hz with an amplitude of 6.65 deg at M=0.9, α=3 deg and $\mathrm{Re} = 17x10^6$ based on the root chord. The computed pressure distributions in terms of magnitude and phase angle are correlated with the experimental data as shown in Figure 2. The prediction is seen to be excellent. The accurate prediction of the phase angle and control surface loads is very important to balance the aircraft. Otherwise inaccurate aircraft state vectors will be computed resulting in wrong design loads and wrong performance characteristics.

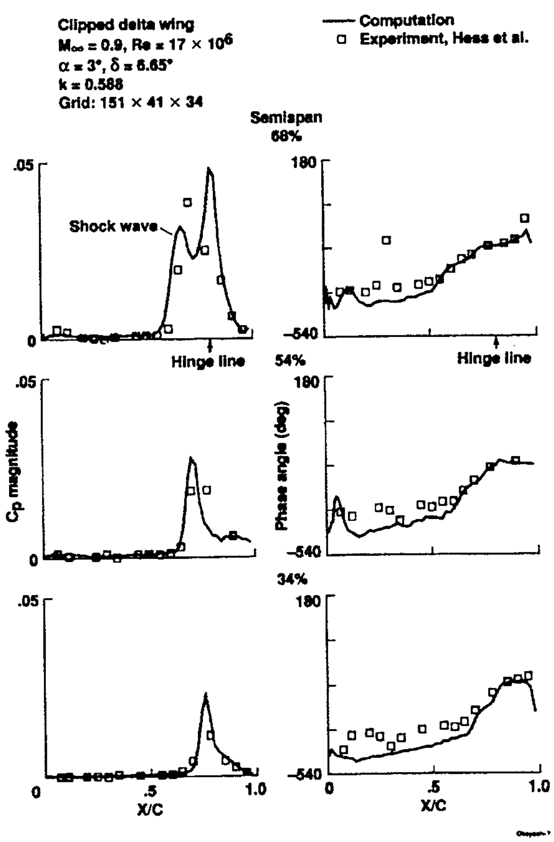

Figure 2. Comparison of unsteady pressure distribution on the upper surface of clipped delta Wing with the experimental data.

REFERENCES

1. Noll, T. E., Austin, E., Donley, S., Graham, G., Harris, T., Kaynes, I., Lee, B. H. K., and Sparrow, J., Impact of Active Controls Technology on Structural Integrity, Journal of Aircraft, Vol. 30, N0. 6, Nov-Dec. 1993.

2. Bhatia, K. G., and Wertheimer, J., Aeroelastic Challenges for a High Speed Civil Transport, AIAA-93-1478-CP, Part 6. The 34th AIAA / ASME /ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, La Jolla, CA. April, 19-22, 1993.

3. Appa, K. and Argyris, J., Computational Aircraft Dynamics and Loads, Sadhana, Vol. 19, Part 3, June, 1994, pp. 467-485, Indian Academy of Sciences, Bangalore, India.

4. Appa, K. and Argyris, J., Nonlinear Multidisciplinary Design Optimization Using System Identification and Optimal Control Theory, Paper AIAA-95-1481, 36th AIAA ASME/ASCE/AHS/ASC, Structures, Structural Dynamics and Materials Conference, April 10-13, 1995/ New Orleans, LA. (also, in Computer Methods in Applied Mechanics and Engineering, Vol. 128, Nos. 3-4/15 December 1995, pp. 419-432.)

5. Guruswamy, G. P. , "User's Guide for ENSAERO - A Multidisciplinary Program for Fluid/Structural/Control Interaction Studies, " NASA TM 108853, October, 1994.

6. Guruswamy, G. P. and Tu, E. L. Euler/Navier-Stokes Flow Computations on Flexible Configurations for Stability Analysis, AIAA 95-1292, 36th AIAA SDM Conf, April 1995, New Orleans, LA

7. Obayashi, S., and Guruswamy, G. P., "Navier-Stokes Computations for Oscillating Control Surfaces," Journal of Aircraft Vol. 31, No. 3, May-June 1994, pp. 631-636.

8. Hess, R. W., Cazier, F. W., Jr., and Wynne, E. C., "Steady and Unsteady Transonic, Pressure Measurements on a Clipped Delta Wing for Pitching and Control-Surface Oscillations," NASA TP-2594, October 1986.